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THE AERODYNAMIC HEATING OF ATMOSPHERE

ENTRY VEHICLES - A REVIEW

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The Aerodynamic Heating of Atmosphere Entry Vehicles -

A Review

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Introduction

It was only a little more than a decade ago that aeronautical engineers were faced with the challenge of designing the first long-range ballistic missiles. Since the chemical rocket requires that oxidizer as well as fuel be carried, the energy content per unit propellant mass is poor. In addition, the mean propulsive efficiency is low so that the ratio of launch weight to empty weight is very large indeed. Hence every effort must be made to keep the payload as large a fraction of the empty weight as possible. Light construction is therefore a prerequisite for the long-range rocket. The engine designers were faced with the problem of designing rocket motors that would produce very high thrust but with little weight. The structural engineers were required to build the rocket shell structure - mainly tankage to hold the large quantity of fuel needed - the mass of which was but a few percent of the total mass at launch. The guidance and control experts were called upon to provide systems which would give a miss distance of about one mile at a target five thousand miles away. The aerodynamicist was asked to provide an entry body for the warhead that could successfully withstand the intense convective heating experienced upon reentry into the Earth's atmosphere. Each of these

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challenging problems - one or two orders of magnitude more difficult than those which had been dealt with previously - was successfully mastered. It is the purpose of this paper to review historically the aerodynamic heating problems of the reentry vehicles starting with long-range ballistic vehicles and finishing with what appear will be the interplanetary space vehicles of the future.

Aerodynamic Heating During Reentry of Ballistic Missiles

A ballistic vehicle required to attain a range of the order of one-quarter the circumference of the Earth must be accelerated during the boost phase into space to a speed of about 7 km/sec. The vehicle then has a kinetic energy of about 25×10^6 m²/sec² per unit mass which is about eight times the amount of energy required to convert a unit mass of ice into steam. Clearly, if a large fraction of this kinetic energy must be converted into heat, only a very small fraction of it can be allowed to heat the vehicle if the vehicle is not to be destroyed before reaching the target. Therein lies the problem.

In order to assess the fraction of the energy which enters the vehicle, consider the following simplified analysis. If the entry speed of the vehicle, V_E , is high and the trajectory reasonably steep, as is the case for the ballistic missile, then the negative acceleration due to drag is large compared to Earth's gravity acceleration during the portion of the atmosphere entry when the heating is important. We may simplify the equation of motion to

$$m \frac{dV}{dt} = - \frac{1}{2} C_D \rho V^2 A \quad (1)$$

where

m entry body mass

V velocity

t time
 ρ air density
 A reference area (usually base area) for definition of coefficients
 C_D drag coefficient

Now, the heating rate for these vehicles is very large compared to the rate at which heat can be reradiated from the surface, and the driving temperature potential promoting the convective transfer of heat is determined essentially by the air temperature (i.e., the wall temperature can be ignored by comparison to the air temperature). Under these circumstances one may show that the rate of input of heat expressed in kinetic energy units is

$$\frac{dq}{dt} = \frac{1}{2} C_H \rho V^3 A \quad (2)$$

where C_H is the dimensionless heat-transfer coefficient.

Then equations (1) and (2) yield the important result that

$$dq = - \frac{C_H}{C_D} \left(\frac{m}{2} dV^2 \right) \quad (3)$$

If we assume for the moment that the mass, the heat-transfer coefficient, and the drag coefficient may be considered constant, then the total heat input is

$$q = \frac{C_H}{C_D} \left[\frac{1}{2} m \left(V_E^2 - V_0^2 \right) \right] \quad (4)$$

where V₀ is the vehicle speed at sea level.

When the total heat input is least it can be shown that the final speed, V₀, is small compared to the entry speed so that approximately

$$q = \frac{C_H}{C_D} \left(\frac{1}{2} m V_E^2 \right) \quad (5)$$

The factor

$$\eta = \frac{C_H}{C_D} \quad (6)$$

is the portion of the total kinetic energy which must appear as heat to the vehicle and $1 - \eta$ the portion which is given to the air. Clearly, we desire η , hereinafter called the energy fraction, to be as small as possible. At ballistic missile speeds the heating process is essentially entirely a convective one. Reynolds analogy (ref. 1) tells us that the convective heat-transfer coefficient is directly expressible as

$$C_H = \frac{1}{2} C_F \quad (7)$$

where C_F is the friction coefficient.

In order to keep the heat shield mass, which is proportional to the total heat input, as small as possible, then, the extraneous mass (see eq. (5)) should be kept as small as possible, so that the vehicle mass m is no larger than necessary. In addition, equations (5) and (7) indicate the optimum will be attained when the ratio of frictional force to total drag force is as small as possible (ref. 2). A blunt body best satisfies this latter requirement. The energy fraction, η , can be made particularly low if laminar flow can be maintained at the usual Reynolds numbers of interest for then the friction coefficient is much less than it would be for turbulent flow. Values of the energy fraction of less than one-half of 1 percent were obtainable so that even using a solid heat sink as a shield - a poor coolant at best - a satisfactory entry vehicle could be designed.

The next important step was to incorporate ablative heat shields in the designs - that is, shields which would vaporize during the heating so that advantage could be taken of the latent heat of vaporization to greatly increase the heat removed per unit of heat shield mass. The ablative shields have a second recognized advantage of great importance (see refs. 3 to 6) which is that the issuing vapor from the shields fends off the air so as to reduce the shear at the vehicle

surface and hence reduce the heat-transfer coefficient itself. This reduction is approximately in the ratio

$$\frac{1}{1 + \left(kV^2 / \zeta_v \right)} \quad (8)$$

wherein

ζ_v the energy required to vaporize a unit mass of the shield

k a constant depending upon the molecular weight of the ablated vapor and upon whether the boundary-layer flow is laminar or turbulent

Values of this constant (ref. 6) are characteristically about 0.3 for laminar flow and about 0.1 for turbulent flow. The important feature to note is that the reduction of heat-transfer coefficient with ablation increases with increasing speed.

There is a third advantage of the ablative shield which is not generally appreciated that will become important at higher entry speeds. It is that as ablation occurs, the coolant, after accepting all the heat of which it is capable, is automatically jettisoned. The ensuing heat load is therefore lessened by the continuous reduction of unnecessary mass. For the ablative heat shield the mass transfer equation is

$$\zeta_v \frac{dm}{dt} = - \frac{1}{2} C_H \rho V^3 A \quad (9)$$

With equation (1) then

$$\frac{dm}{m} = \frac{C_H V}{C_D \zeta_v} dV = \frac{C_H}{2C_D \zeta_v} dV^2 \quad (10)$$

so that for constant heat-transfer coefficient and drag coefficient, the final mass is, for a low speed at impact,

$$\frac{m_f}{m_E} = e^{-\frac{C_H V E^2}{2C_D \zeta_v}} \approx \frac{m_p}{m_E} \quad (11)$$

since the final mass is, of course, the payload in an optimum design. For the heat sink shield, on the other hand, even supposing the coolant is as efficient as the ablator and that no allowance is made for the favorable effect of ablation on the heat-transfer coefficient, the optimum ratio of payload to total mass of payload plus coolant is

$$\frac{m_p}{m_E} = 1 - \frac{C_H V_E^2}{2C_D \zeta v} \quad (12)$$

Figure 1 shows the optimum ratio of payload to entry mass as a function of the energy parameter $C_H V_E^2 / 2C_D \zeta v$. For the nonablative shield the payload vanishes when the energy parameter reaches 1. For the ablative shield some payload is available for indefinitely large values of the energy parameter although it may be uneconomically small.

Reentry Heating for Space Vehicles

The negative accelerations of ballistic vehicles entering the atmosphere on steep trajectories are large compared to the acceleration of gravity, as noted earlier, so that the equation of motion can be approximated by equation (1) and the trajectory is essentially a straight line. In planetary atmospheres the density variation with altitude is essentially exponential in form

$$\rho = \rho_0 e^{-\beta h} \quad (13)$$

where

ρ_0 sea-level density

h altitude

β inverse of the scale height

Under these circumstances it may readily be shown (ref. 2) that the maximum acceleration, if it is reached before impact, is

$$\frac{dv}{dt_{\max}} = - \frac{\beta V_E^2 \sin \gamma}{2e} \quad (14)$$

where

γ angle between the trajectory and the local horizontal

e Napierian base

This is a rather curious result since it indicates this acceleration to be independent of the vehicle shape or mass and only dependent on entry speed, trajectory angle, and the scale height of the atmosphere. For the typical ballistic missile the maximum accelerations can reach about $-60 g$ but this is of small significance since such vehicles can be rather easily made sufficiently robust to resist such loads.

For manned space vehicles, of course, maximum accelerations must be limited to values of the order of $10 g$ during entry. Equation (14) gives the clue that this can be accomplished with a shallow trajectory. This equation, however, cannot be used to find the permissible trajectory angles because gravity effects were ignored in its formulation. Chapman (refs. 7 and 8) has considered this problem with gravity effects included as well as the effects of the use of aerodynamic lift. For near-Earth manned satellites, such as for Project Mercury, a satisfactory reentry may be made without the use of aerodynamic lift if the entry trajectory is neither so shallow as to overly extend the trajectory due to insufficient drag nor so steep as to promote excessive accelerations. In fact, by the use of a modest retrorocket with properly directed thrust a landing at a preselected spot can easily be effected without the use of aerodynamic lift, as has been demonstrated with the Mercury spacecraft. When the entry speed is increased to values corresponding to Earth parabolic speed or greater, the attainment of a permissible approach trajectory becomes more difficult. At Earth hyperbolic speeds, in fact, if the approach is too shallow the drag may not be sufficient to assure that the vehicle will be "captured" by the atmosphere. Generally, the

situation will be more restricted than has been indicated, for if the vehicle on the shallow trajectory leaves the atmosphere, it may traverse the Van Allen belts before its next approach to Earth and so subject the occupants to a lethal radiation hazard. Accordingly, it usually is assumed that a manned vehicle must be captured and kept in the atmosphere during the "first pass." A concept convenient to discussion of the manned entry vehicle problem is the so-called entry corridor defined herein as the range of altitudes required as aiming points for the approach to assure that the vehicle neither experiences excessive accelerations nor fails to be captured in a single pass. Figure 2 shows the 10 g corridor as a function of lift-drag ratio for entry at Earth parabolic speed. The advantage of using aerodynamic lift is apparent and the subject has been treated extensively in the literature (e.g., refs. 7 to 11).

This digression into the discussion of trajectory requirements is pertinent to the heating problem. For small rugged entry vehicles of rather low mass, steep trajectories are generally preferable for minimizing ablative heat-shield weight, if aerodynamic loading is not a factor, since the heat pulse though intense is but of very short duration so that little heat is conducted into the substructure. In addition, ablative shields which melt before being vaporized have a thin melt so that the flow of the melt layer creates little difficulty. In contrast, for manned vehicles which must employ shallow-angled trajectories to avoid excessive accelerations, the heating rate is more modest but lasts for a considerable period. The conductivity problem is so severe that insulation is required to prevent overheating of the substructure. The choice of ablative materials is restricted since many

will flow unduly because the liquid film layer is relatively thick. On the other hand, there are some compensating factors favoring the shallow trajectories: The long heating time permits a sizable amount of heat to be radiated from the vehicle surface, thus reducing the required mass for the ablator. Also for large heavy vehicles the flight Reynolds numbers are lower so that a laminar flow can often be maintained where otherwise turbulent flow would occur, hence the heating rate is lessened. The use of varying aerodynamic lift complicates the problem since the surface distribution of heating rate varies with the vehicle attitude.

Up to this point the tacit assumption has been made that convective heating constitutes the total. For speeds up to nearly parabolic speed for Earth, this is essentially the case. Thus the total heat input, other things being equal, only increases as about the square of the speed. It is not immediately apparent that there is a necessity to consider entry speeds above parabolic speed for Earth return from travel to the neighboring planets. For Hohmann transfer trajectories from Mars or Venus to Earth, the atmosphere entry speeds at Earth are essentially Earth parabolic speed. However, as shown in figure 3, the times required for the minimum energy trips are long, substantial fractions of a year. There are many good reasons for wanting to shorten the travel time. For the occupants, long flight duration will probably promote some difficult psychological problems. Certainly, too, the weight of life support equipment increases with voyage time. More obscurely, the vehicle weight, as determined for equal meteor impact hazard, for example, gives advantage to short trip duration. Thus, for even modest sojourns into space there are valid reasons for considering

atmosphere entry speeds in excess of the parabolic speed. For more distant journeys of the future the demands are even greater.

At these higher speeds one must contend with an additional heating contribution which arises in the following way: The air entering the shock layer is drastically slowed down relative to the body and is highly compressed. The high kinetic energy is then almost entirely converted to heat. This excitation at the higher flight speeds is sufficient to dissociate and ionize a large fraction of the compressed gas. These atomic and molecular species become important sources of radiation which serve to promote additional heating of the vehicle (see, e.g., refs. 12 to 16).

The phenomenon is a complicated one since a chain of processes is required to establish chemical and thermodynamic equilibrium. Thus the radiation from the shock layer varies along the stream lines as air initially out of equilibrium subsides to the equilibrium state. For the purposes of this discussion it is sufficient to note that one can regard the shock-layer radiation as having two components, one from that fraction of the gas which is in equilibrium and one from the nonequilibrium fraction. When radiative heating becomes comparable to or exceeds the convective heating, it has been found that the equilibrium component far overshadows the nonequilibrium component. Accordingly, we shall concentrate our attention on the equilibrium radiation herein.

Figure 4 shows the equilibrium radiation rate per unit shock-layer gas volume as a function of U , the upstream velocity component normal to the shock for several values of flight altitude. These are calculated characteristics (shown by the symbolized points) and the lines composed of straight segments are arbitrary fairings. At a given velocity U , the

variation of radiation with altitude corresponds to a variation with air density to the 1.8 power. At any given altitude on increasing the velocity, U , from about 8 km/sec to 13.7 km/sec, the intensity is increased four orders of magnitude, the intensity varying with the velocity to the 15.45 power. Above the speed of 13.7 km/sec, the radiation continues to increase but only as the velocity to the 5.05 power. It is readily apparent that although radiative heating constitutes a trivial contribution at the lower speed, it becomes the dominating factor at higher speeds, particularly if these high speeds are attained at low altitudes.

One concludes that when atmosphere entry speeds exceed about Earth parabolic speed (11 km/sec) the blunt-body solution is no longer the optimum. Conical shapes for vehicles become attractive in this higher speed range since the bow shock essentially is no longer normal to the direction of motion. As we have seen for normal shock speeds up to nearly 14 km/sec, the principal radiative contribution varies as something more than the fifteenth power of the velocity component normal to the shock. Thus, for a given flight speed, the radiative rate varies as the fifteenth power of the sine of the bow shock angle, and so the radiative input is drastically reduced for even very modest inclination of the bow shock. The conical shape is less favorable than the blunt body from the convective heating aspect, but since the required cone shape is not one of high fineness ratio, the penalty is small. This is readily apparent in the next three figures, taken from reference 17 which contains an extended analysis of the heating of conical entry vehicles at speeds in excess of Earth parabolic speed. Figure 5 shows the fraction of the entry kinetic energy of the vehicle which must be

contended with as heat to the vehicle as a function of entry speed for a conical shaped body having a half-cone angle of 30° of arc. The ablator assumed here is Teflon, and the dimensionless ballistic parameter

$$B = \frac{C_D \rho_o A}{\beta m \sin \gamma} \quad (15)$$

has a value of 200 which, as estimated from our meager present knowledge, is about as small as can be allowed if the flow, as assumed for this case, is to be laminar. Note, here, that the radiative heating does not contribute substantially until speeds of the order of 20 kilometers are exceeded after which it dominates. The nonequilibrium radiative contribution η_n , as noted earlier, remains small compared to the equilibrium radiative contribution η_e and the laminar convective portion, η_l .

Figure 6 shows the total energy fraction as a function of entry speed for a range of cone half-angles for the same value of the ballistic parameter. The optimum energy fractions for this value of B are shown by the envelope (dotted curve). These values are but a small part of 1 percent of the entry kinetic energy.

If one assumes that laminar flow can be maintained up to a Reynolds number of 10^7 based on body length and local flow characteristics, the fraction of the entry mass which must be ablated as a function of entry speed for two ablators (subliming Teflon and vaporizing quartz) is shown in figure 7. It is seen that the ablated mass can apparently be kept a reasonably small fraction of the entry mass even to fairly high entry speed - well into the range of entry speeds of meteors. If turbulent flow occurs, the mass loss fraction is much greater since the energy fraction is increased about an order of magnitude. Considerable future

research on the age-old transition problem is clearly needed at these high speeds. We must learn the factor favorable to the maintenance of laminar flow in boundary layers composed of air and ablation vapors.

There is another problem with conically shaped entry vehicles to touch on before we leave this subject. The convective heat transfer is highest at the cone apex and diminishes toward the cone base. Thus the cone ablates to a round-nosed, near-cone shape with increased cone half-angle as the atmosphere entry progresses. If the entry speed is high, the rounded nose becomes flattened and extended in width by the radiative heating contribution at the lower altitudes. Much of the advantage of the original conical shape may thus be lost. In reference 17 this problem is treated in some detail and it appears that the penalty can be kept small, provided excessive shape change is avoided by extra film cooling at the apex. Of course, the mechanical complication attendant with this cooling is undesirable but it is a price which perhaps must be paid to effect a satisfactory solution to the problem.

For manned vehicles at these higher speeds the problem of providing an adequate altitude corridor becomes severe. As shown in figure 8, for an assumed fixed maximum deceleration of 10 g, a higher and higher lift-drag ratio is required as entry speed increases until a speed is reached for which the corridor vanishes. For this deceleration the limit speed is about 26 km/sec, this being the speed for which, with zero drag, the centrifugal force experienced is 10 g for a flight trajectory having a curvature equal to the radius of the Earth. The required aerodynamic lift force is, of course, directed toward the

Earth's center. From the foregoing it appears that the entry speeds for manned vehicles, at least, will have to be limited to values less than perhaps 20 km/sec and that lift must be provided. Entry vehicles shaped like half-cones, perhaps, may provide satisfactory configurations (ref. 10) for high-speed entries.

Up to this point it has been tacitly assumed that aerodynamic braking of a vehicle is preferable to rocket braking to effect a landing on Earth. It is well to digress, here, to make a comparison. As is well known (see ref. 18) the specific impulse of a rocket is defined as

$$i = \frac{T}{g_0(dm/dt)} \quad (16)$$

where

i specific impulse, sec

T thrust, kg m/sec²

g_0 gravitational acceleration at Earth surface, m/sec²

dm/dt time rate of mass flow of rocket propellants

For rocket braking this thrust provides the negative acceleration

$$\frac{dV}{dt} = - \frac{T}{m} \quad (17)$$

Equations (16) and (17) can be combined to give

$$\frac{dm}{m} = - \frac{dV}{g_0 i} \quad (18)$$

Comparing this equation with the corresponding one for the ablation rate with aerodynamic braking (eq. (10)), one sees that the equivalent specific impulse of an aerodynamically braked vehicle using an ablative heat shield is

$$i_{eq} = \frac{C_D \zeta_V}{C_H g_0 V} = \frac{\zeta_V}{\eta g_0 V} \quad (19)$$

where η is the energy fraction

For the optimum conical vehicles of reference 17 with laminar boundary-layer flow, and assuming

$$\zeta_v = 2.2 \times 10^6 \text{ m}^2/\text{sec}^2$$

for subliming Teflon and

$$\zeta_v = 16 \times 10^6 \text{ m}^2/\text{sec}^2$$

for vaporizing quartz, one can calculate the equivalent specific impulse for aerodynamic braking. The variation of this impulse with speed for these two assumed ablators is shown in figure 9. Though these equivalent impulses drop with increasing entry speed, they are always well above what can be attained with chemical rockets (less than 500 sec) or nuclear rockets (up to about 1,000 sec) even for a rather inefficient ablator such as Teflon. Hence as long as energy fractions for vehicles with ablative heat shields can be kept to the order of 1 percent or so, rocket braking cannot be considered competitive.

Aerodynamic Braking in the Atmospheres of Venus and Mars

The Earth's close neighbors, Venus and Mars, are objects of particular interest in the space age. Although the atmospheres of these planets have been the object of study for many years, the facts are few. Both atmospheres have carbon dioxide as an important constituent. The estimates range from a few percent to about half of the total. Nitrogen, presumably, and perhaps Argon are the other principal constituents. Apparently oxygen and water vapor are present in but relatively small amounts. In any study of atmosphere entry heating for these planets we must, for the present, assume that the relative amounts of the constituents are varied over a wide range.

The Venus atmosphere is assuredly much more dense than is the Earth's so that atmospheric braking is not difficult. On the other hand,

the temperature near the surface of the planet is high so that even an instrument package may not survive for long. The Mars atmosphere is tenuous. Recent observations (ref. 19) indicate a very low surface pressure indeed (25 ± 15 millibars) so that braking an entry vehicle to a low speed at surface impact may be impossible without the use of a retrorocket at touchdown. The atmospheric temperature near the surface is apparently even less than for Earth, so vehicle survival after landing does not appear to present a problem.

Convective heat transfer in the Venus and Mars atmosphere appears to present no problem. Comparison of convective heat in CO_2 and air (ref. 20) shows only a minor difference which is in essential agreement with theory (ref. 21). On the other hand, the radiative heating contributed from the shock layer is generally a more serious problem than for air (ref. 22) because of the formation of cyanogen, a strong radiator, from the nitrogen and carbon dioxide constituents. At speeds of 6 or 7 km/sec the experimental evidence indicates that the radiative heating is about one order of magnitude greater than for air. However, as speed is increased, the $\text{CO}_2 - \text{N}_2$ mixtures approach more nearly air values of radiative heating. My colleague, James Arnold, has recently measured the effects of adding argon and has found, contrary to expectation, that this diluent does not appreciably influence the basic $\text{CO}_2 - \text{N}_2$ results.

Because the actual composition of the Venus and Mars' atmospheres is doubtful, these estimates of the severity of the heating problem may be in considerable error.

Experimental Determination of Aerodynamic Heating at Hypervelocities

The accuracy of the foregoing reentry heating analyses is only as good as the accuracy of our basic knowledge of the chemical and thermodynamic processes involved. As in the scientific delineation of all natural philosophy, theory is vitally important to our understanding of the heating phenomena in high-enthalpy air flows. As is often the case, constant input from experimental research is needed not only to progress at a fast pace but to assure that the theoretical results are, in fact, valid. In hypersonic heat-transfer phenomena this is particularly true because many of the basic physical concepts involved are not too well understood. Unfortunately, the present upper limit on air speed in our ground-based laboratory equipment is only a little more than 13 km/sec. For present analyses at higher air speeds, we must rely on extrapolation by theory. In time we expect the limit of attainable speeds in the laboratory will increase but by unknown amounts. It is possible to perform flight tests using rockets to extend our experimental knowledge to the higher speeds, but such tests are very expensive and so time consuming that they cannot be counted on to produce the experimental data we need. One then looks for other experimental sources for confirmation.

Observations of meteor flight are one such source worthy of consideration. Meteoroids are known to enter the Earth's atmosphere at speeds ranging from parabolic speed (11 km/sec) up to the highest speed they can have and still be members of the solar system (72 km/sec). The entry speed requirements for our purposes are, therefore, more than met. On the other hand, the atmosphere entry of meteoroids large enough to enjoy continuum flow are sufficiently infrequent that only occasionally

can one be tracked at any one location. Until recently there have existed but a few meteor observatories so that the number of continuum flow meteors, for which accurate tracking data are available, is very limited. As will be discussed more fully later, there are presently under construction many new meteor observatories so that future prospects as regards the availability of meteor data are good.

Astronomers, for a number of years now (see ref. 23 for a review), have successfully tracked meteors in the following way: One or more cameras, located at the two ends of a known base line, are provided with rotating shutters which occult the meteor image on the photographic plates at even time intervals. The meteor trajectory is determined by triangulation. The variation of velocity and, in turn, acceleration, of the meteoric body along the trajectory is determined from the interrupted photographic track. Comparison of relative exposure of the meteor track with the background field of stars whose photographic magnitudes are known provides the measurement of the variation of the meteor luminosity along the trajectory. Spectra are often measured to assess the meteor composition. These data are sufficient, in principle, to allow the determination of the variation of size and heat-transfer characteristics of the meteoric body with altitude and speed. Two methods of analysis generally employed are the "dynamical method" and the "photometrical method." We shall treat only the former in this paper. The latter is discussed in reference 24. Suffice it to say, here, that the methods are essentially redundant if meteor density is known. When the density is unknown, the two methods are needed for the complete solution.

In the dynamical method of analysis the velocity and acceleration histories are all that are needed to determine heat-transfer characteristics, and, if the density of the meteoroid is known, the size variation with altitude can be found as well. However, it is presumed that air-density variation with altitude, h , and the body shape are known. It is usual to employ a standard atmosphere (e.g., ref. 25) for air density values. The body is assumed to remain essentially spherical during the atmosphere entry since a sphere represents about the best mean of possible shapes (see ref. 26). From the equation of motion (eq. (1)) the product of the meteor density and radius as a function of altitude can be found from

$$\rho_m r = - \frac{3C_D \rho_0}{8} \left(\frac{\bar{\rho} V^2}{dV/dt} \right) \quad (20)$$

where $\bar{\rho} = \rho/\rho_0$ and for the assumed spherical shape, C_D is approximately unity. If ρ_m is known, then the size can be determined. Equations (1) and (9) combine to give the heat-transfer parameter

$$\frac{C_H}{C_D \zeta} = \frac{3}{\bar{\rho} V^3} \frac{d}{dt} \left(\frac{\bar{\rho} V^2}{dV/dt} \right) = - \frac{3 \sin \gamma}{\bar{\rho} V^2} \frac{d}{dh} \left(\frac{\bar{\rho} V^2}{dV/dt} \right) \quad (21)$$

since

$$dt = - \frac{dh}{V \sin \gamma} \quad (22)$$

Here we note a major difficulty - that even presuming C_D is known, we cannot find C_H unless the heat of ablation is known, which it is not. There is, however, an upper limit to the possible value of ζ which is the total energy required to bring the surface material from the cold state to the vapor state so that an upper limit for C_H can be found in any event. For stone meteoroids one expects, a priori, that vaporization would be the usual ablation process since liquid stone is

rather viscous - but if the entry speed is low or the trajectory is not steep, then the mass ablated in the liquid state (as Chapman, refs. 27 and 28, has found for tektites) can be a rather large fraction of the total. On the other hand, for large meteoroids which enter the atmosphere on a steep trajectory and at high speeds liquid run-off would be small. However, an important fraction of the ablated mass may be in the solid state since stone is a weak material, and, because the thermal conductivity is low, it may also spall as the result of excessive thermal stress. For iron meteoroids one does not expect ablation to be important in the solid state except for bodies of great size since this material is strong and the thermal conductivity is relatively high. However liquid iron has a very low viscosity so that ablation principally as a liquid should be expected.

A serious weakness of the dynamical method is that in passing from velocity to meteor size to the heat-transfer factor each function involves, in turn, differentiation of the last. Hence, unless the velocity is very accurately defined as a function of time and altitude, the final results may be subject to large mean error. The analysis (see ref. 29) of the record of the Canadian meteor "Meanook 132" indicates that velocities can, in fact, be determined with about the required accuracy. Table I gives the velocity and acceleration history for this meteor. These results (see ref. 24) are more complete than the original values given in reference 29 and include a correction to one of the acceleration values (private communication from Dr. Millman). These data are plotted in figures 10 and 11 along with the curves determined by a sixth-degree least squares fit to all the velocity and acceleration data. The close agreement between the data and the curves

indicates that the velocity and acceleration data are self-consistent. In figure 12 the dotted curve shows the variation of the meteor density-radius product as computed from the sixth-order least squares series. The "point-to-point" values are calculated from each of the individual velocities and accelerations. The solid curve gives the photometrical results for a density of 850 kg/m^3 .

Figure 13 gives the corresponding variation of the heat-transfer parameter with altitude. The dotted curve, again, is obtained from the sixth-degree least squares fit while the circled points are from neighboring point values of velocity and of acceleration to evaluate the mean value over each altitude interval. These dynamical results compare favorably with the photometrical results, as the solid line shows. The dash-dot curves give estimated values of the heat-transfer parameter when in one case only vapor ablation was assumed to occur and in the other only fluid ablation. The implication, here, is that considerable ablation in the solid state must have occurred. This low-density material (less than water density) must evidently be porous and, accordingly, weak so that solid ablation would be expected. Hence, the results for this meteor do not add to our knowledge of the heating of typical man-made entry vehicles but they do show that the meteor data derived from the photographic plates must be of high quality.

The data for the Sacramento Peak Meteor 19816 (ref. 30) analyzed in reference 24 do yield results which are encouraging, for they indicate that extrapolation of our present knowledge of heat transfer up to speeds of about 20 km/sec may not be far from fact. The velocity and accelerations with the corresponding least squares errors (indicated by ΔV and $\Delta dV/dt$) are given in table II for this meteor whose spectrum indicates that its composition is typical of asteroidal stone ($\rho_m = 3400 \text{ kg/m}^3$).

The results of the dynamical analyses (point-to-point and least squares series) are shown in figure 14 along with the values estimated when vapor and fluid ablation are assumed. Even when only vapor ablation is assumed, the data indicate values of the heat-transfer coefficient only about twice as high as those estimated. As noted earlier some fluid ablation probably occurred for this small stone (the entry radius is about 1 cm). A proper estimate would be somewhat increased therefore so that the data and a proper estimate would be in better agreement.

It is probable, in fact, that at these higher speeds our estimates of heat-transfer coefficient are low because one additional source of heating has not been accounted for. The air molecules in the boundary layer and in the wake collide with the ablated vapor molecules. When these collisions are sufficiently energetic, as for the air-to-air molecular collisions in the shock layer, they become a source of radiation. The observed luminosity of meteors is, in large part (or completely in free molecular flow), a result of this radiation. The photometrical method of analysis for meteors employs, in fact, this radiation from the ablated vapor collisions as a means for determining the mass loss rate of a meteoric body. The magnitude of the radiation per unit ablated mass depends not only on the flight speed (i.e., collision energy) but also on the composition of the vapors. Even at rather low speeds the radiation can be important for certain ablaters. Figure 15 is a photograph of a Lexan model in flight. The photo was taken with an image-converter camera by my colleague Max Wilkins at Ames Research Center. The 1/2-inch-diameter model, which has a round nosed cone for the forward face, is in flight in a ballistic range at a speed of 7.2 km/sec. All of the wake radiation and a large part of

the radiation from the region of the forward face results from "ablation products radiation." Some important features of this radiation are that, in addition to heating the forward face, it may greatly augment the afterbody heating of high-speed reentry bodies. Also, as noted by Craig and Davy (ref. 31, see also ref. 24) this radiative heating tends to become self-perpetuating at sufficiently high speeds. Designers of future high-speed entry vehicles would do well to carefully choose ablators for heat shields which have low ablation-product radiation.

To return to the subject of meteors, it does appear that meteor tracking records can provide experimental heat-transfer characteristics at very high entry speeds. In particular, we stand to learn a good deal about ablation-products radiation and about the ability of various ablative materials to resist structural failure caused by thermal stress, provided the meteor composition can be determined. Recently the NASA supported a proposal by the Smithsonian Astrophysical Observatory to construct and operate a network of meteor observatories covering a large area in the midwest. This "Prairie Network" is intended to provide tracking data on bright fireballs with sufficient precision to allow the meteorites to be retrieved. Improved analysis of these tracking records should result since, for the retrieved bodies, the final mass and shape and the meteor density and composition will be known. Also, as noted earlier, many more tracking records for meteors in continuum flow should become available than have been available up to the present.

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TABLE I.- FLIGHT DATA FOR MEANOOK METEOR 132

(sin γ = 0.868)

t, sec	h, km	V, km/sec	dV/dt, km/sec ²
0	67.59	17.42	-0.31
.1		17.39	-.36
.2	64.56	17.35	-.42
.3		17.30	-.50
.4	61.55	17.25	-.60
.5		17.18	-.73
.6	58.57	17.10	-.88
.7		17.00	-1.06
.8	55.58	16.89	-1.28
.9		16.75	-1.55
1.0	52.71	16.58	-1.88
1.1		16.37	-2.28
1.2	49.87	16.12	-2.76
1.3		15.82	-3.34
1.4	47.12	15.45	-4.02
1.5		15.01	-4.77
1.6	44.51	14.49	-5.67
1.7		13.86	-6.97
1.8	42.11	13.10	-8.18
1.9		12.24	-9.08
2.0	39.98	11.28	-9.97
2.1		10.25	-10.58
2.2	38.21	9.16	-11.21
2.3	37.46	(8.00)	

TABLE II.- FLIGHT DATA FOR SACRAMENTO PEAK METEOR 19816

(sin γ = 0.716)

t, sec	h, km	V, km/sec	ΔV , km/sec	dV/dt, km/sec ²	$\Delta dV/dt$, km/sec ²
0.764	80.13	20.67	± 0.01	-0.158	± 0.009
1.135	74.02	20.57	± 0.02	-.416	± 0.018
1.860	64.16	19.87	± 0.05	-2.02	± 0.05
2.038	61.06	19.27	± 0.09	-2.75	± 0.10
2.425	56.49	17.85	± 0.11	-5.92	± 0.11
2.623	53.75	16.36	± 0.18	-8.45	± 0.18
2.824	51.87	14.63	± 0.19	-9.67	± 0.19
3.124	48.92	10.77	± 0.30	-8.90	± 0.29

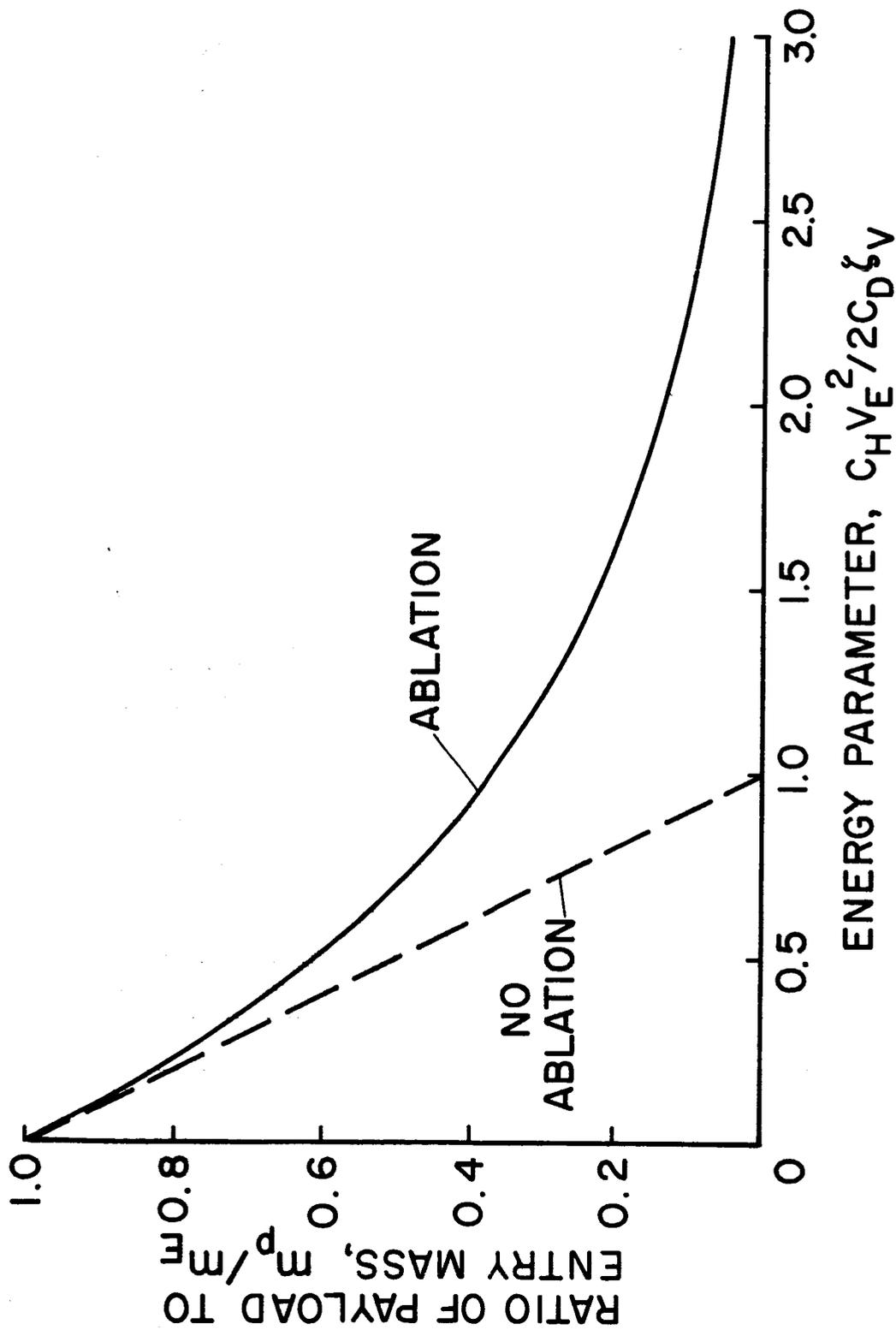


Figure 1.- Optimum payload fraction for ablative and nonablative heat shields.

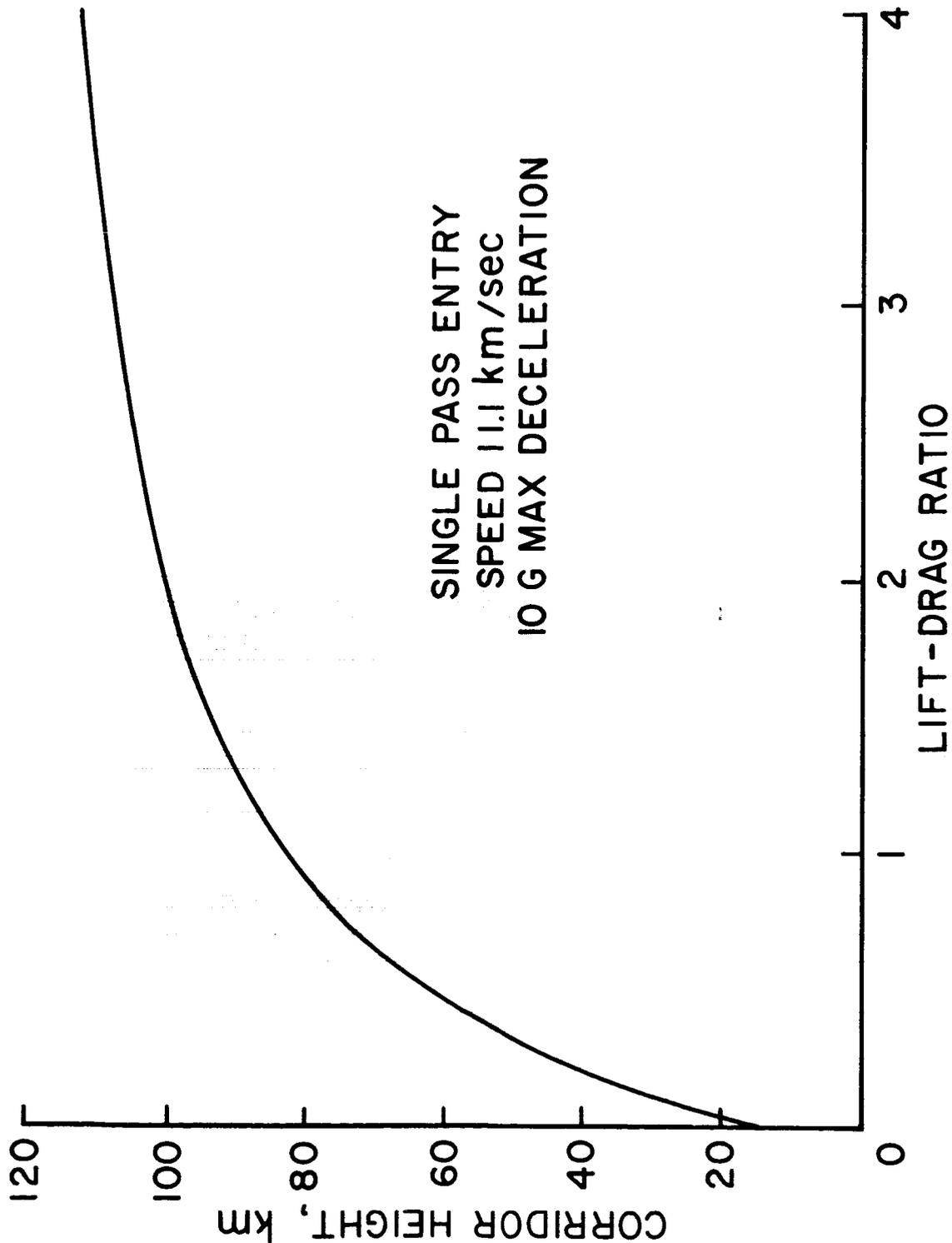


Figure 2.- Effect of lift-drag ratio on corridor height at earth parabolic speed.

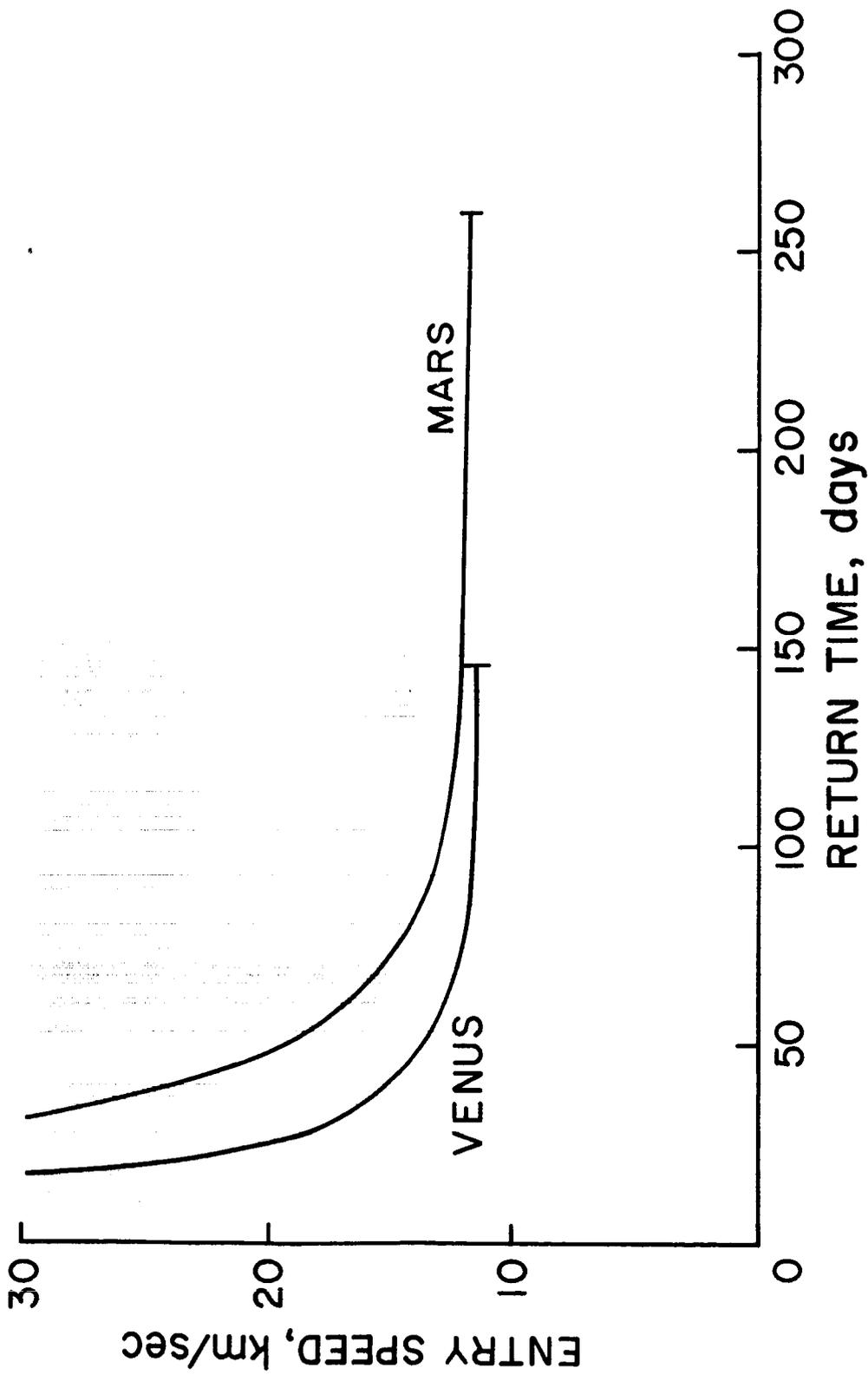


Figure 3.- Earth entry speed as a function of transit time from Mars and Venus.

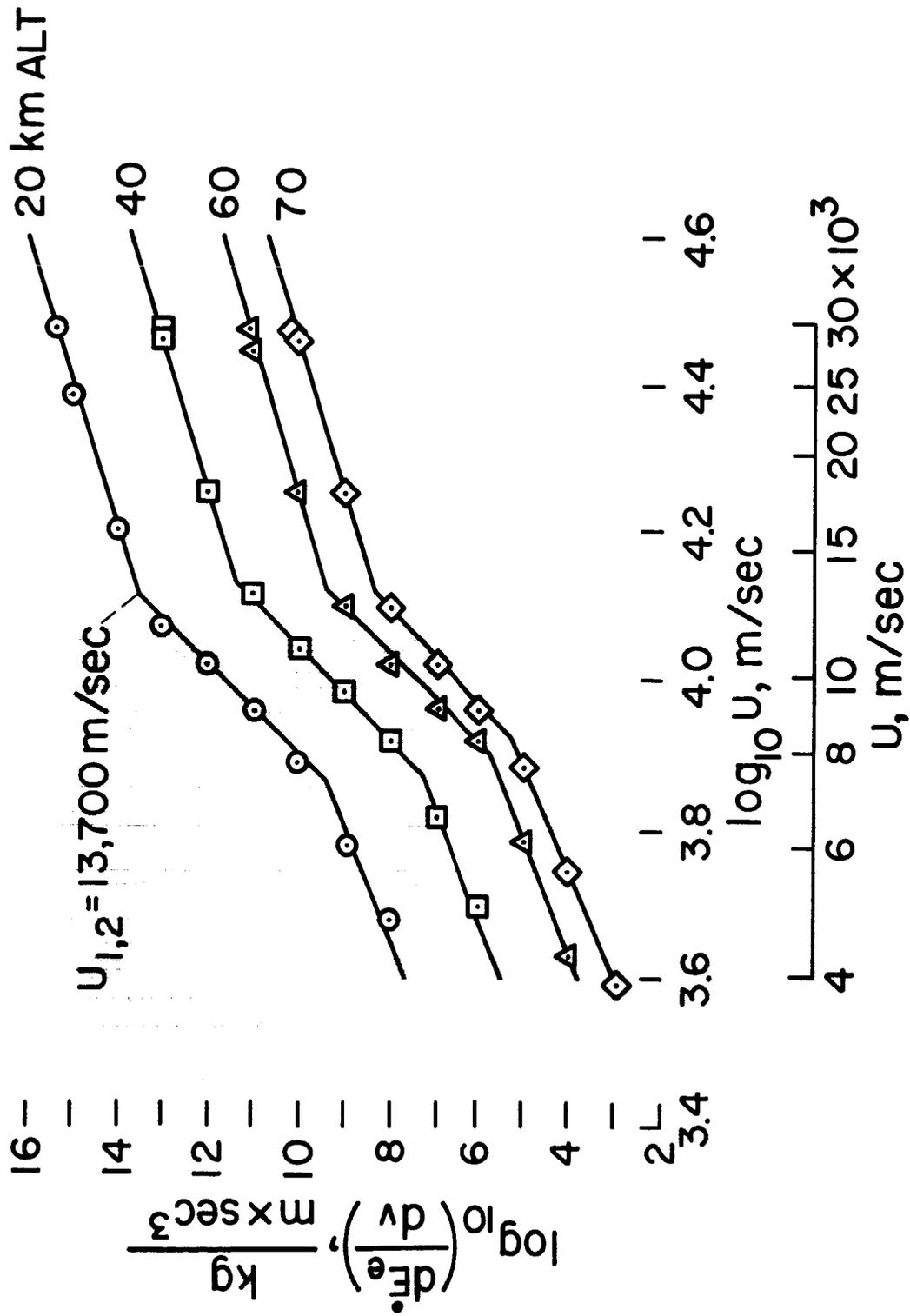


Figure 4.- Equilibrium radiation from shock layer.

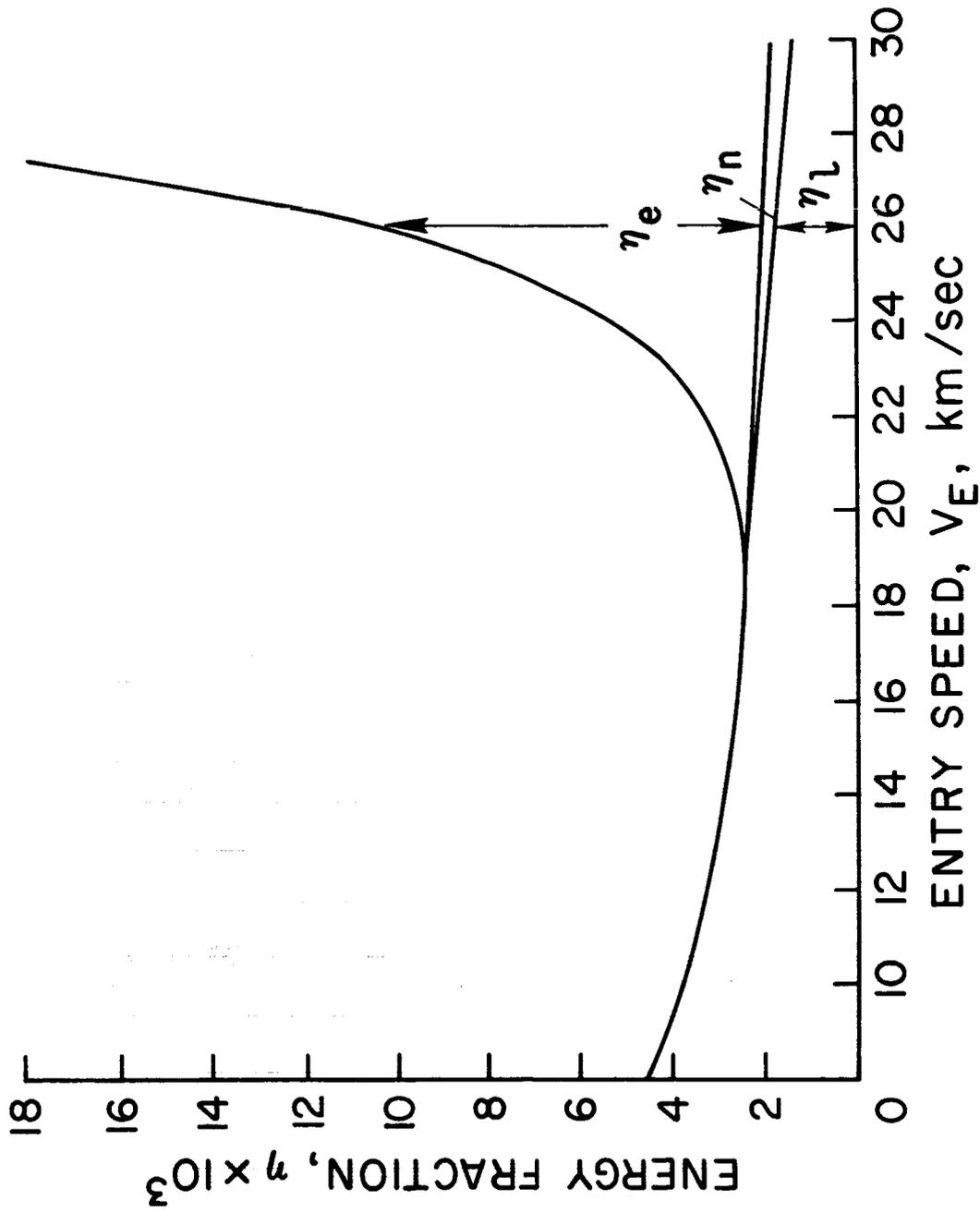


Figure 5.- Energy fractions for cones with $\theta_c = 30^\circ$, $B = 200$.

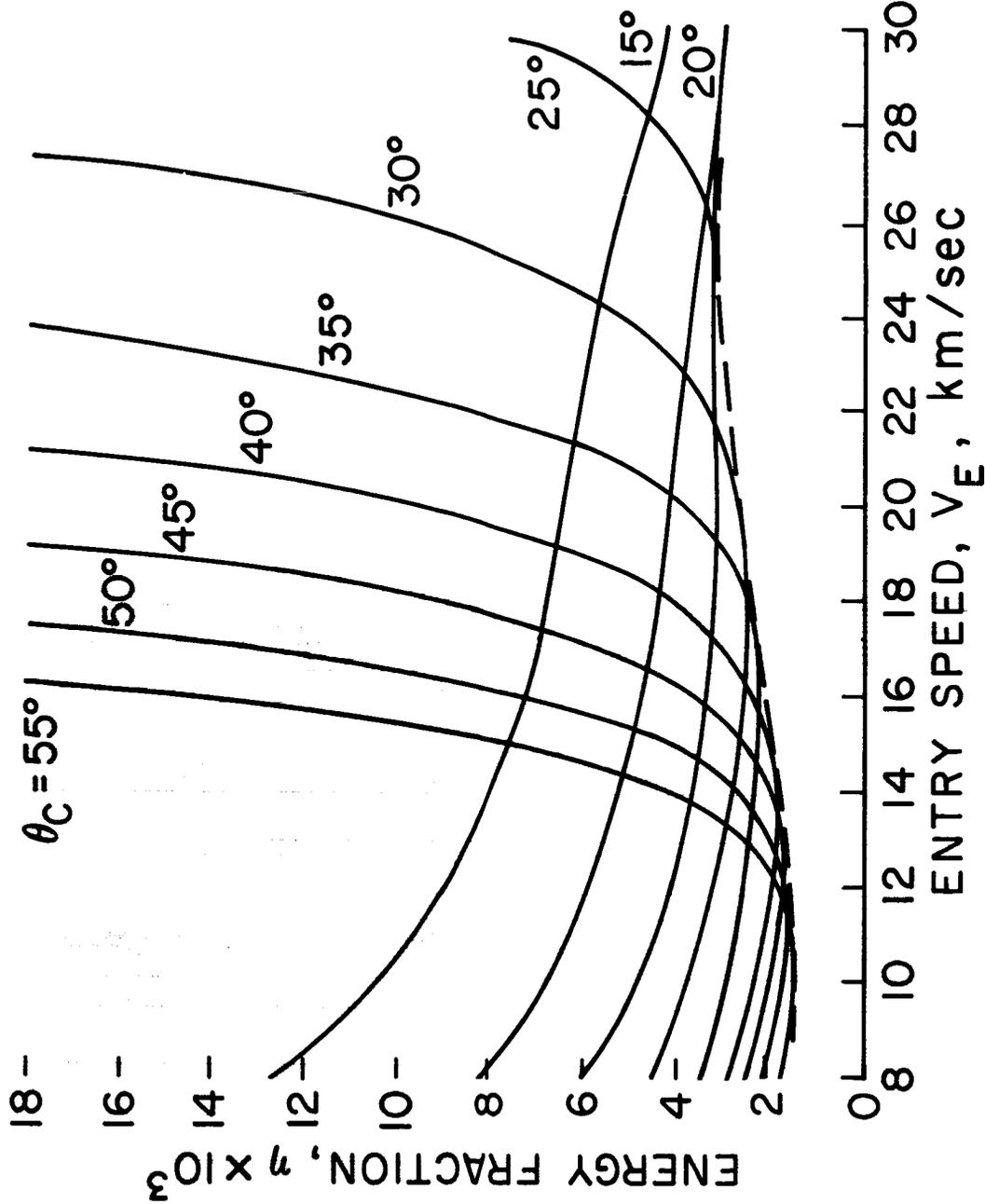


Figure 6.- Energy fractions for cones with $B = 200$.

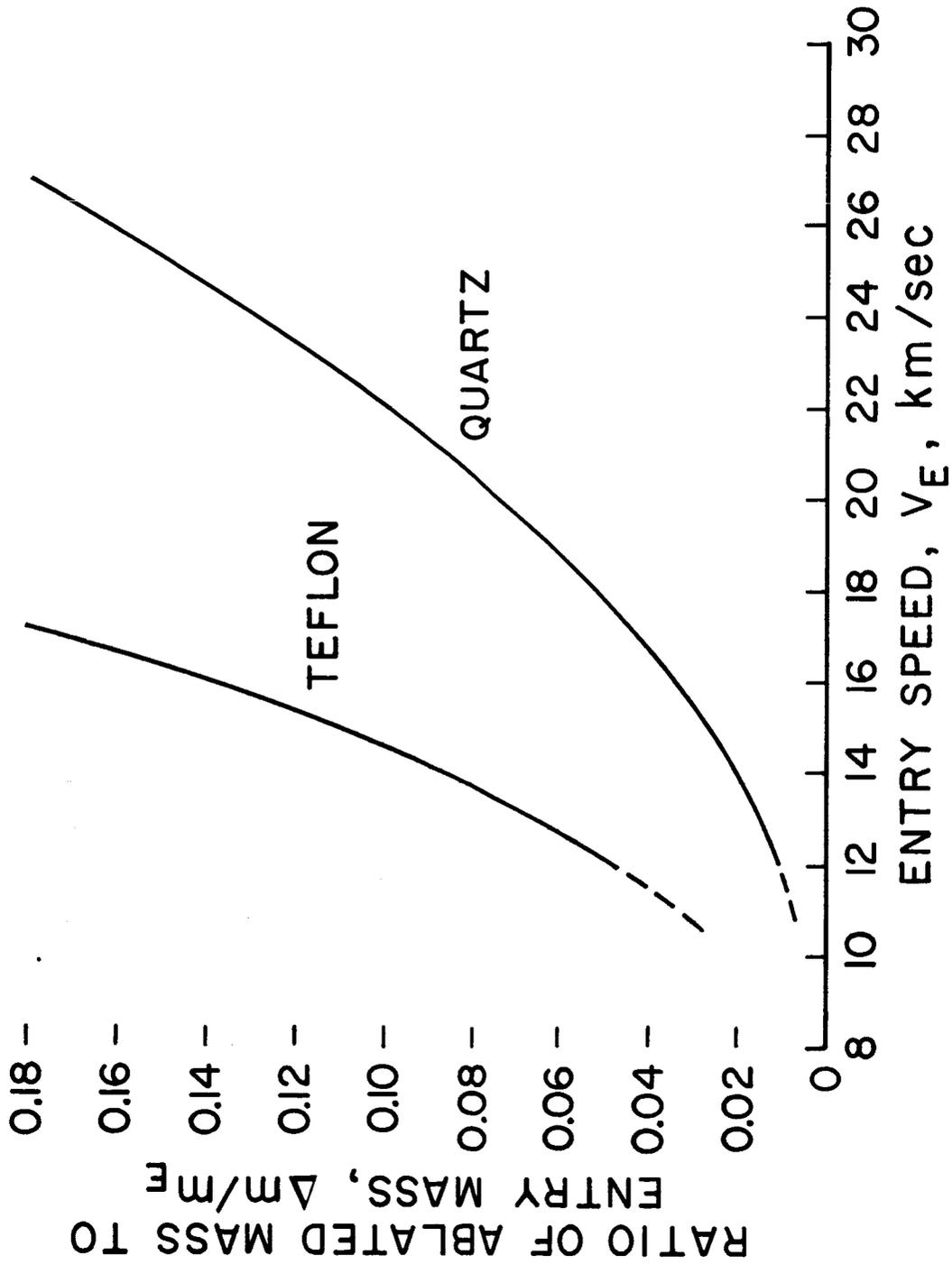


Figure 7.- Optimum mass-loss ratios for two ablators.

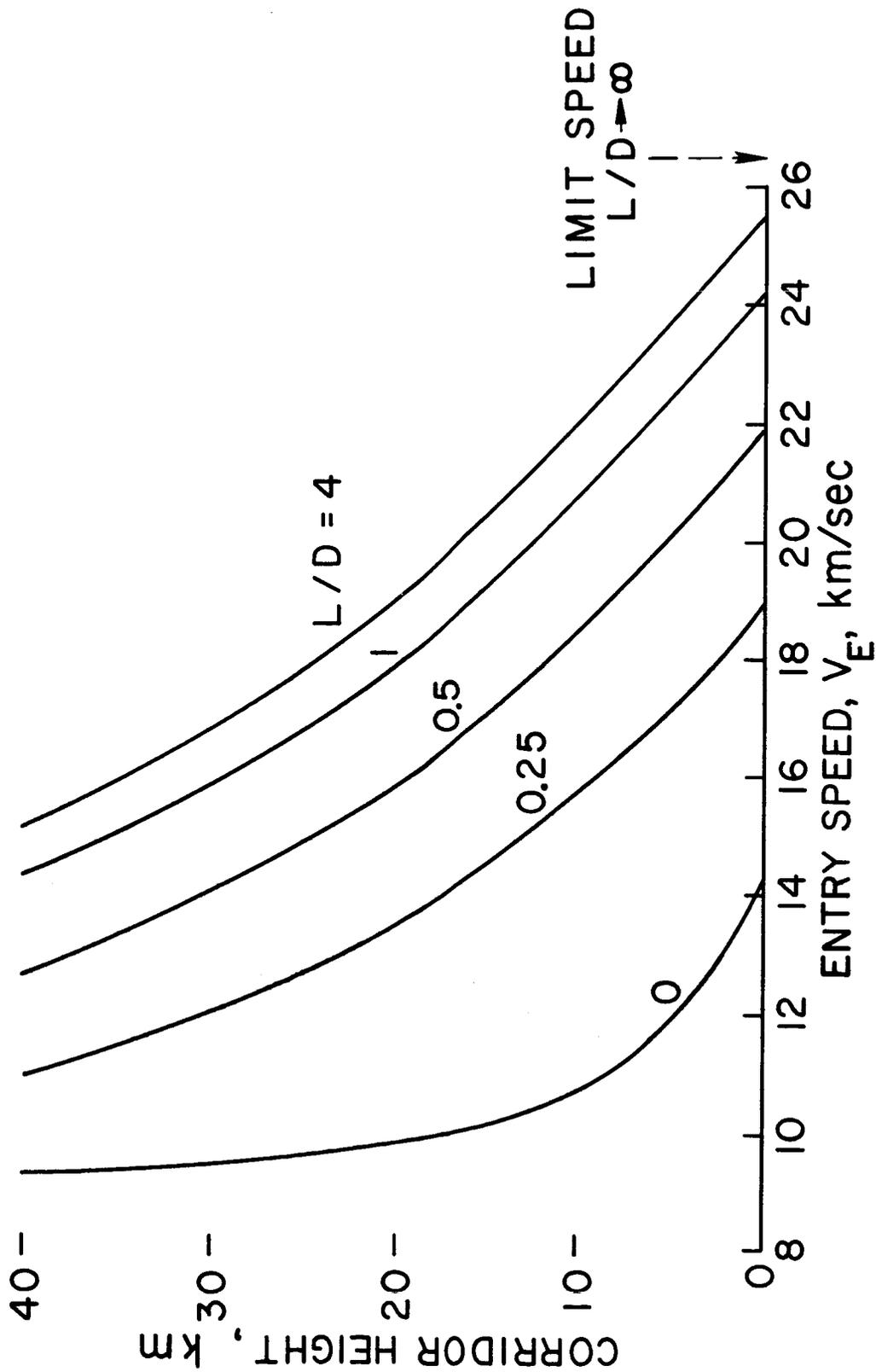


Figure 8.- Entry corridor height as a function of entry speed and vehicle lift-drag ratio.

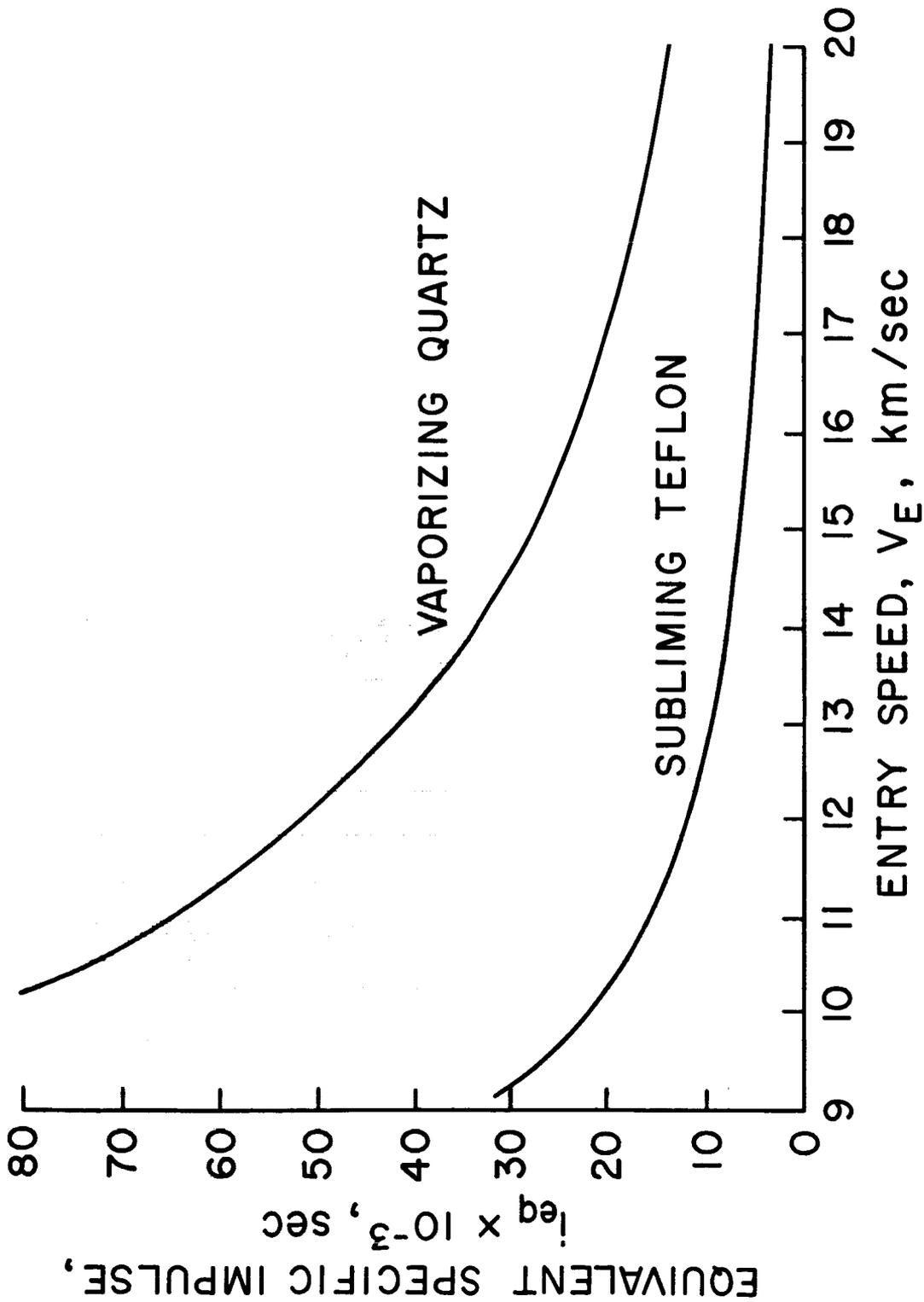


Figure 9.- Equivalent specific impulse of aerodynamically braked entry vehicle.

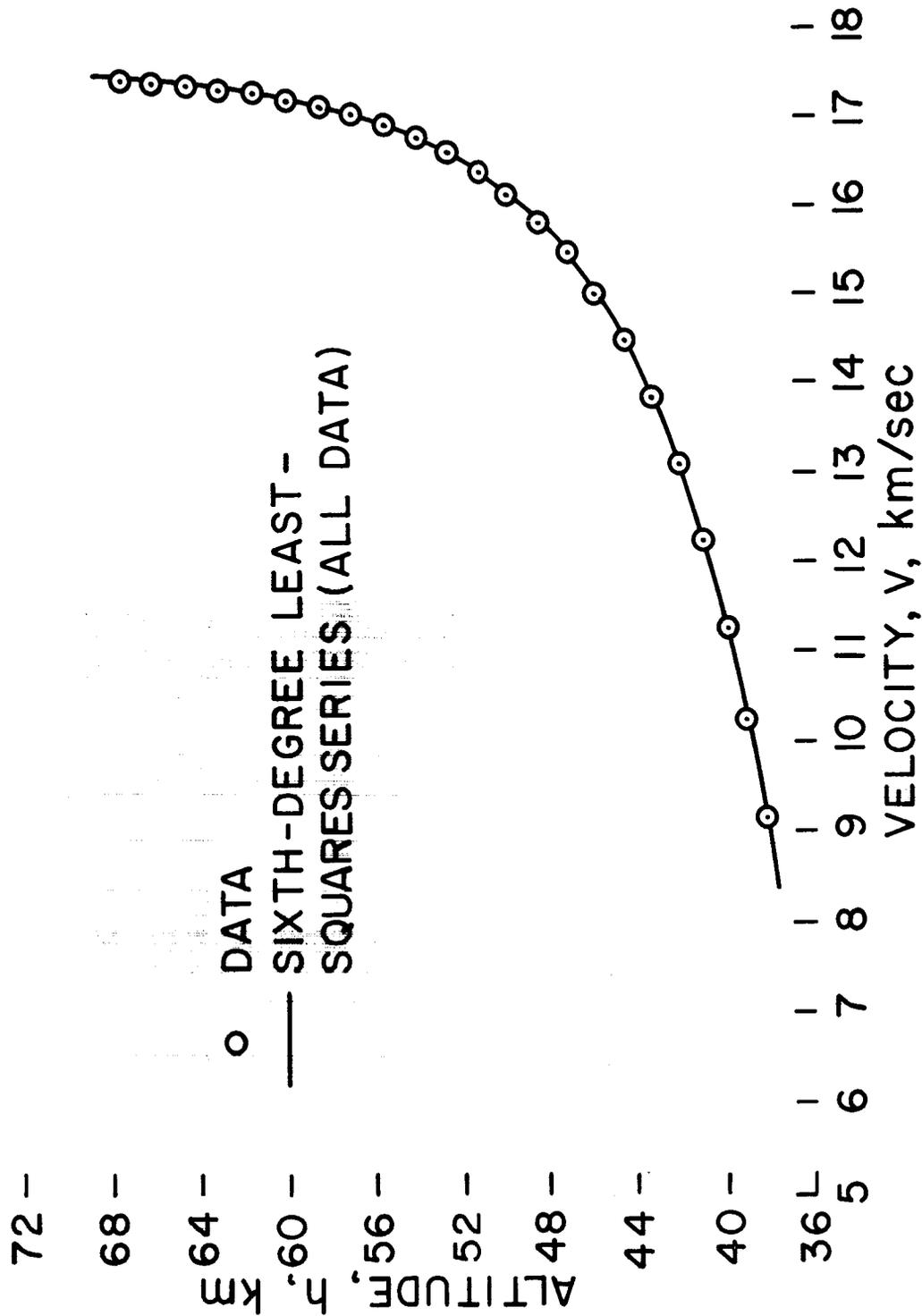


Figure 10.- Meanook Meteor 132.

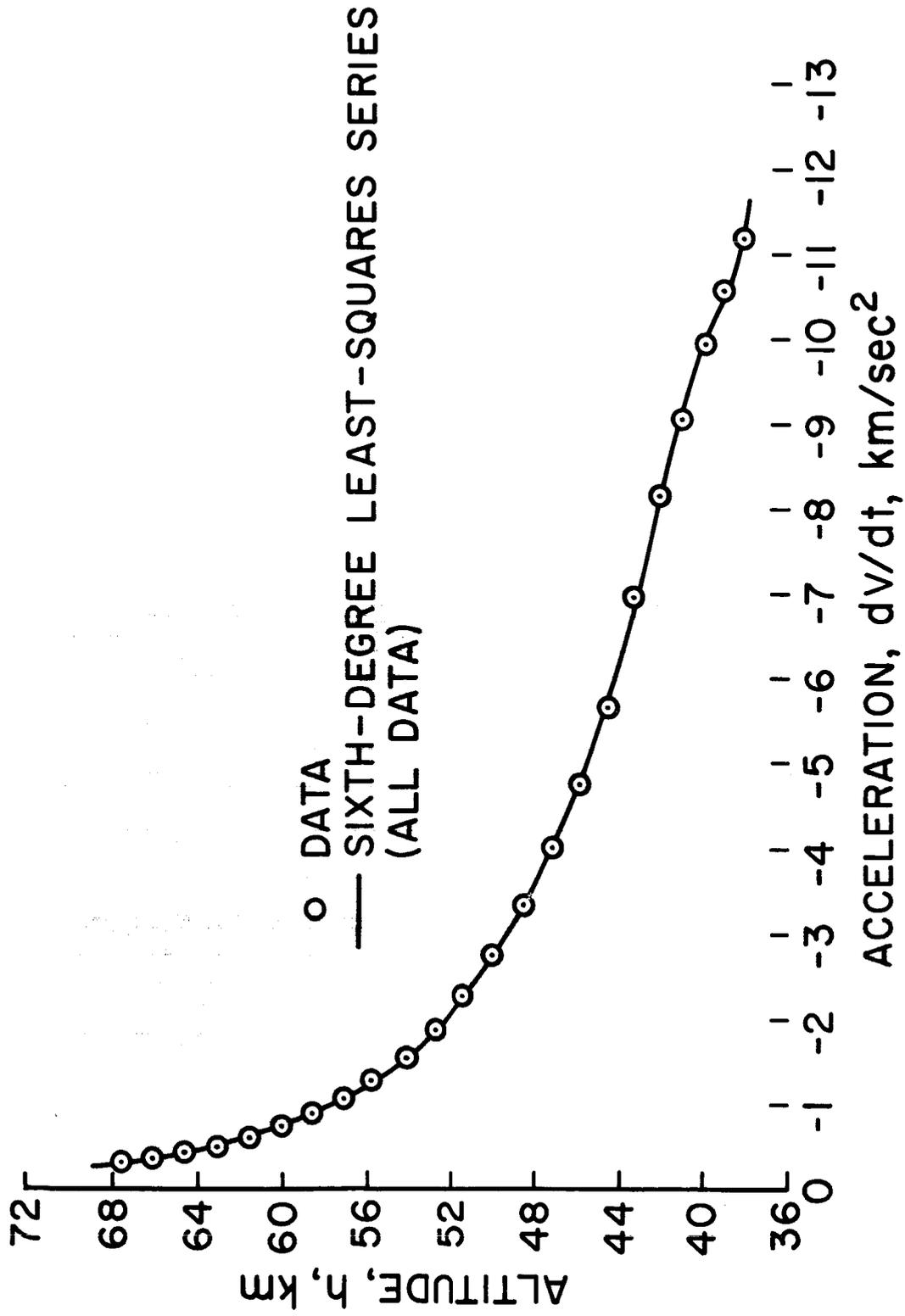


Figure 11.- Meanook Meteor 132.

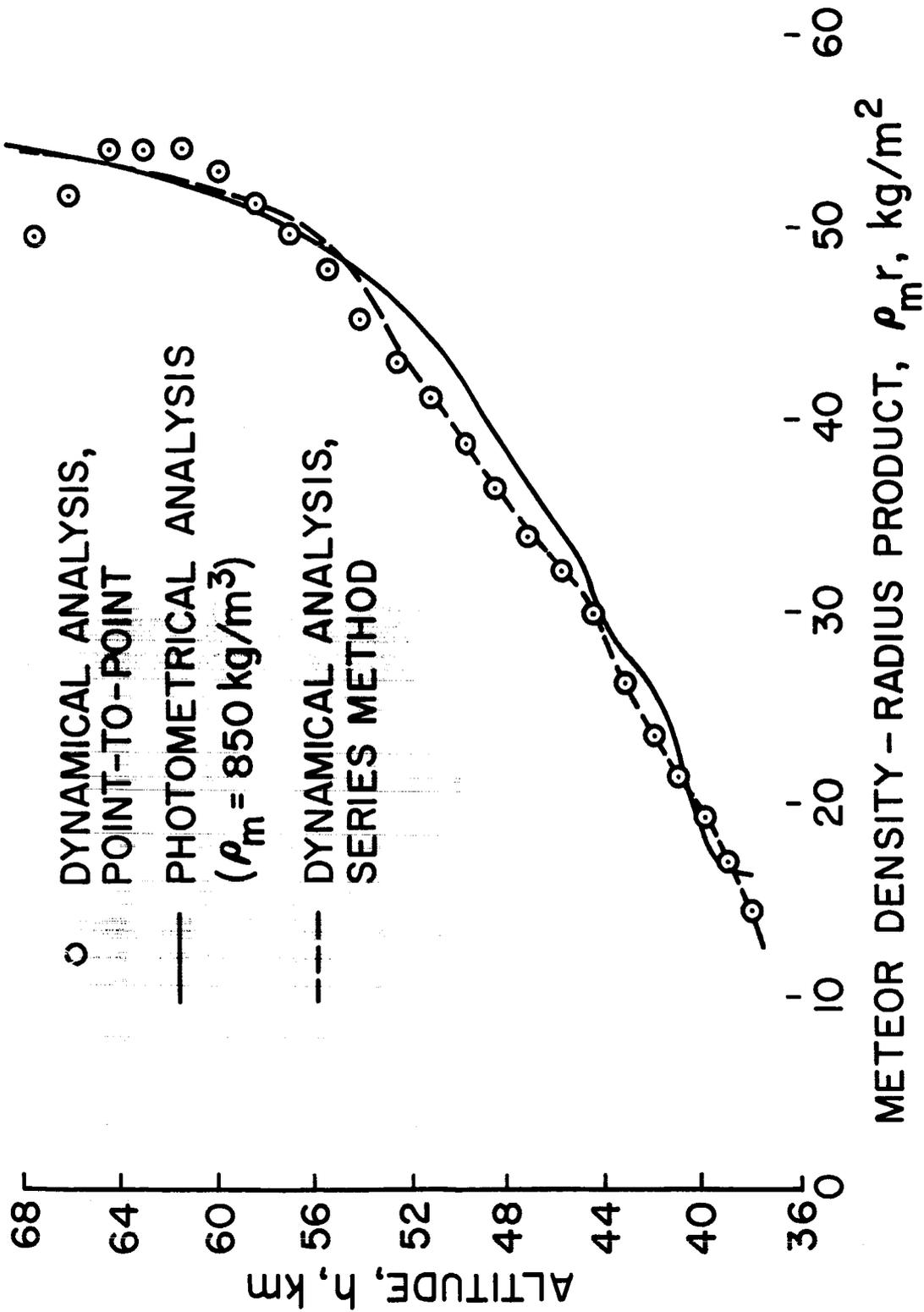


Figure 12.-- Meanook Meteor 132.

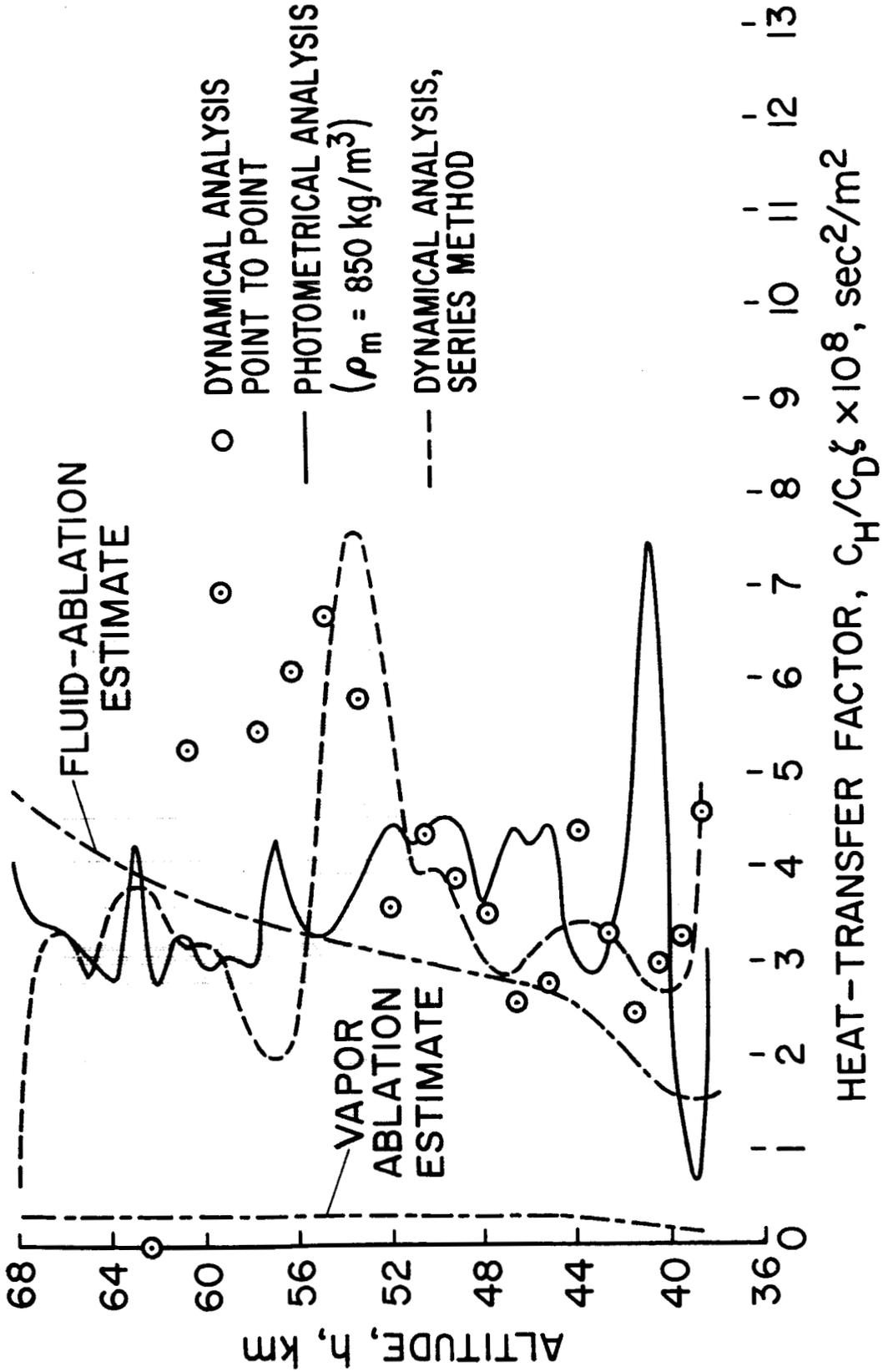


Figure 13.- Meanook Meteor 132.

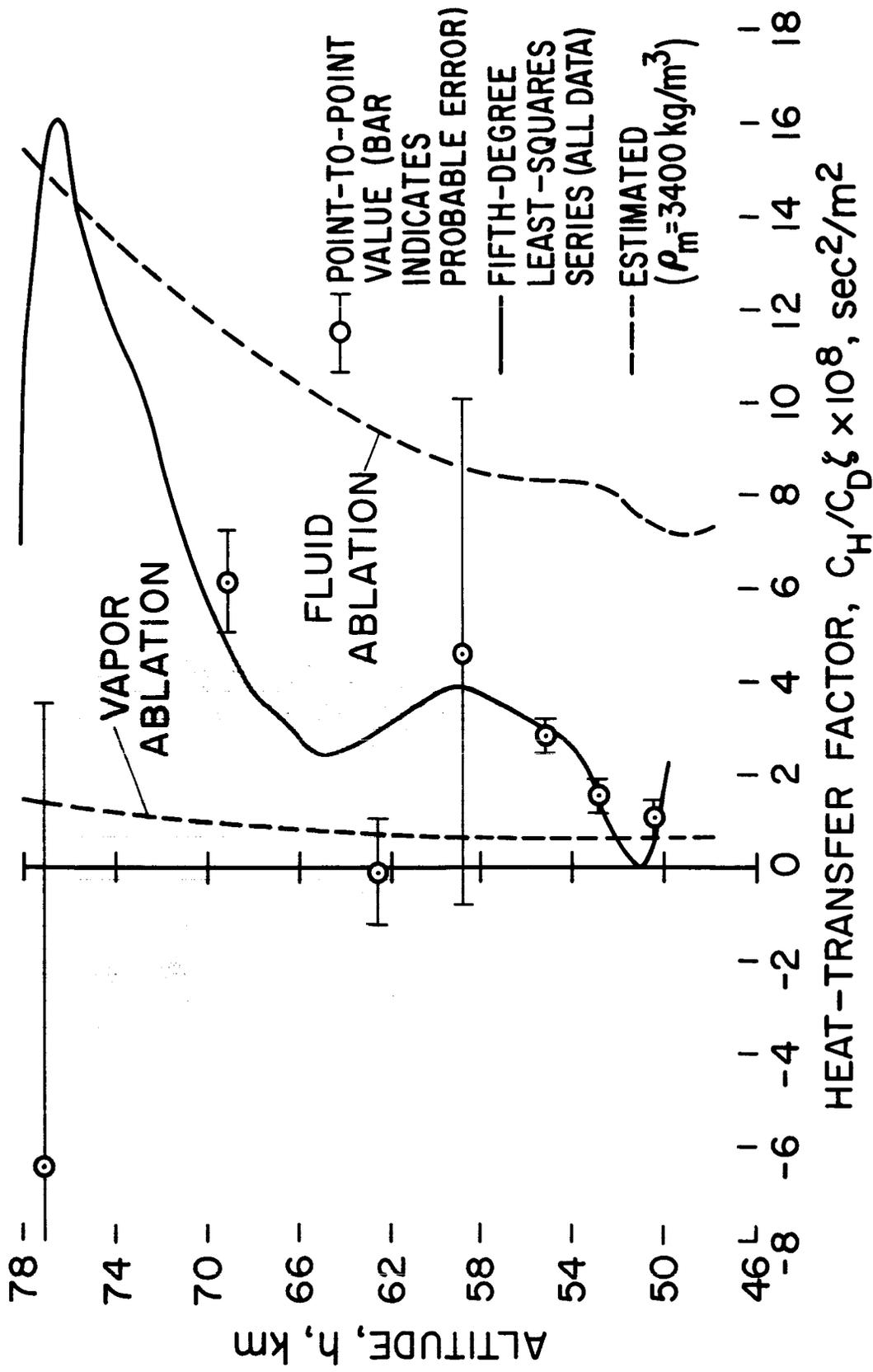


Figure 14.- Sacramento Peak Meteor 19816.

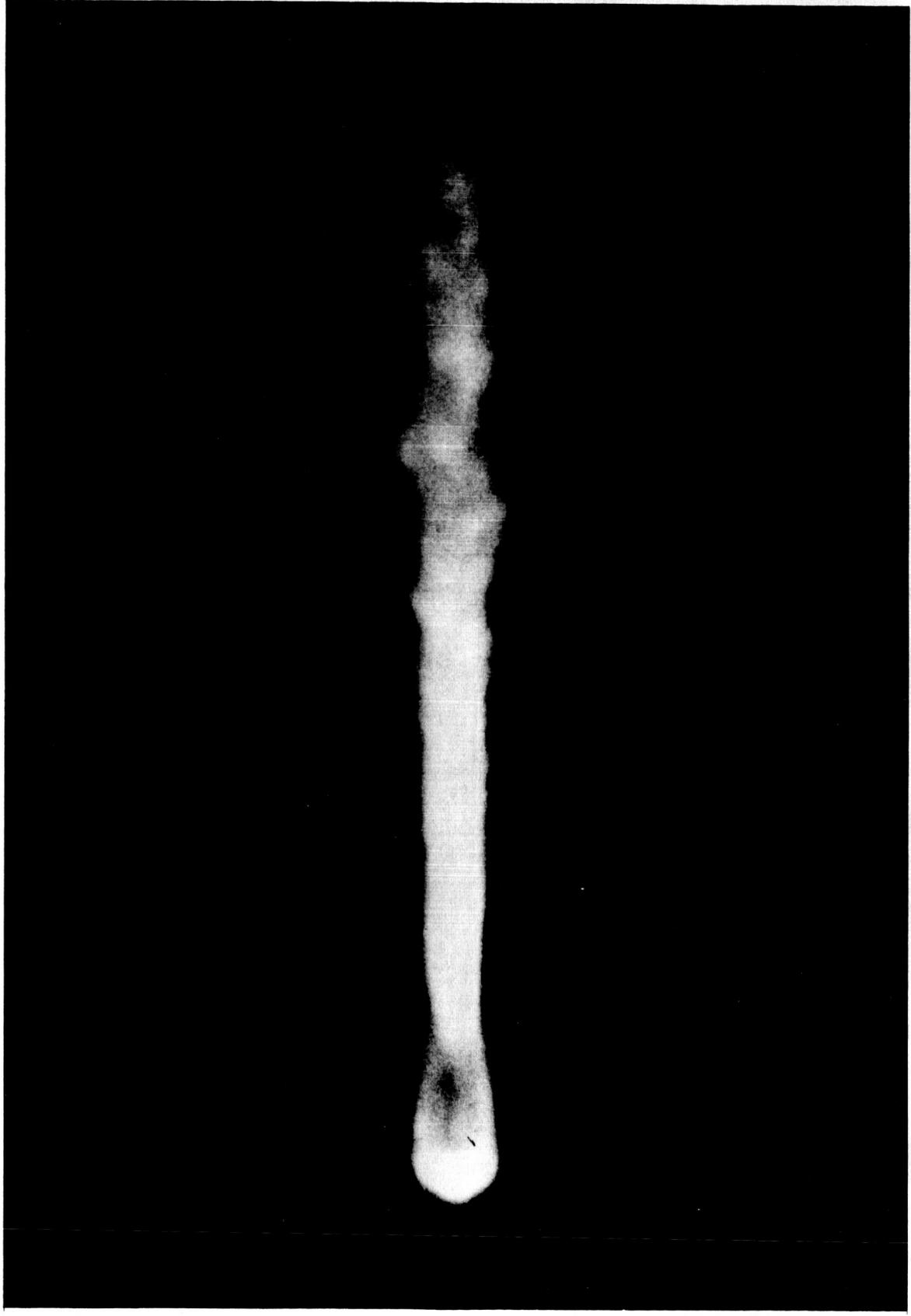


Figure 15.- A laboratory-simulated meteor.

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